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THOR DELTA DESCRIPTION AND PAYLOAD CAPABILITY

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MISSILE & SPACE SYSTEMS DIVISION
DOUGLAS AIRCRAFT COMPANY, INC.
SANTA MONICA/CALIFORNIA

DOUGLAS

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DOUGLAS AIRCRAFT COMPANY, INC

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1.0 INTRODUCTION AND SUMMARY

This document presents the Thor/Delta payload capabilities for North Polar circular and elliptical orbits and for space probes.

Proposed launch sites and range safety requirements should be reviewed by the United States Government in light of Thor/Delta experience in the United States.

A description of the Thor/Delta vehicle and performance characteristics pertaining to it are also given.

2. DESCRIPTION OF THE THOR/DELTA VEHICLE

Thor/Delta is a fully developed and flight proven system. Its reliability record in accomplishment of orbital missions and diverse space probes is unequalled.

Figures 1 and 2 present outboard and inboard profiles respectively of the Delta Vehicle.

2.1 First Stage Vehicle

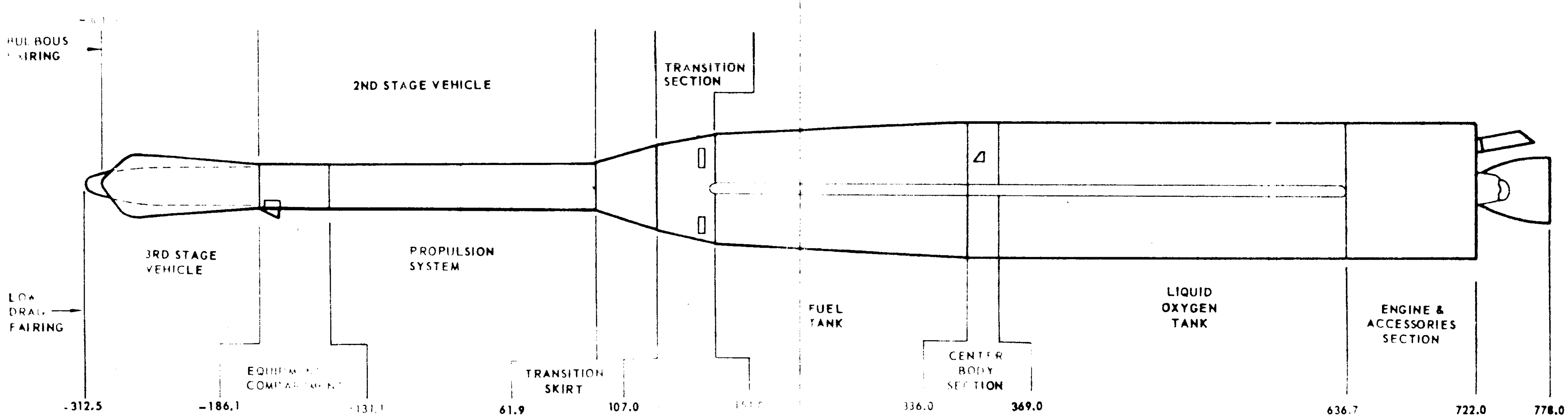
2.1.1 General Description

The first stage is a Model DM-21 Thor vehicle (figure 3) consisting of a transition section, fuel and liquid oxygen tanks, a center body section, and an engine and accessories section. In addition, the transition section structure is modified to accommodate the loads imposed by the second and third stage vehicles.

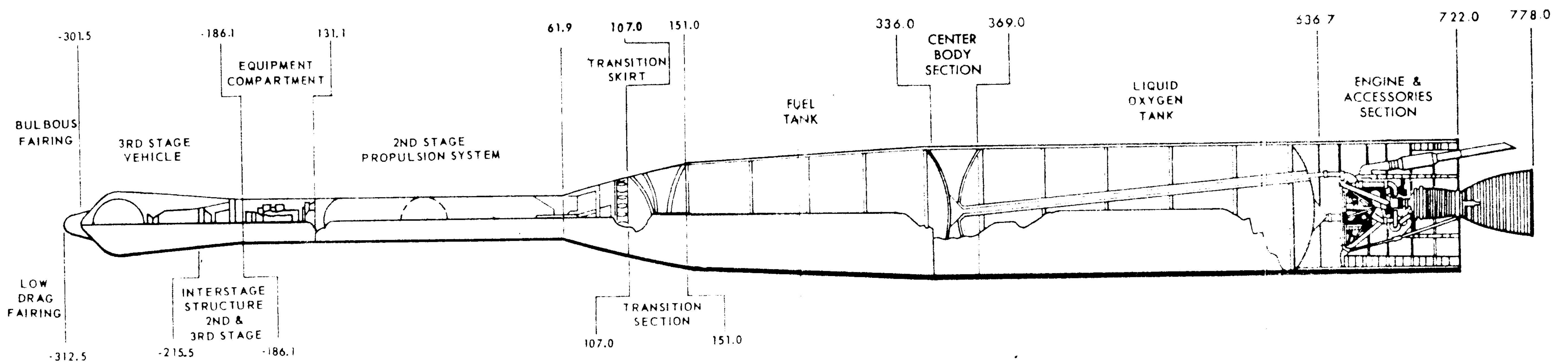
2.1.2 Propulsion System

The Delta booster vehicle propulsion system consists of a Rocketdyne MB-3 Block II main engine, rated at 170,000 pounds thrust (sea level stabilized) and two Rocketdyne vernier engines, rated at 1,000 pounds thrust (sea level) per engine. Liquid oxygen and RJ-1 fuel are used as propellants.

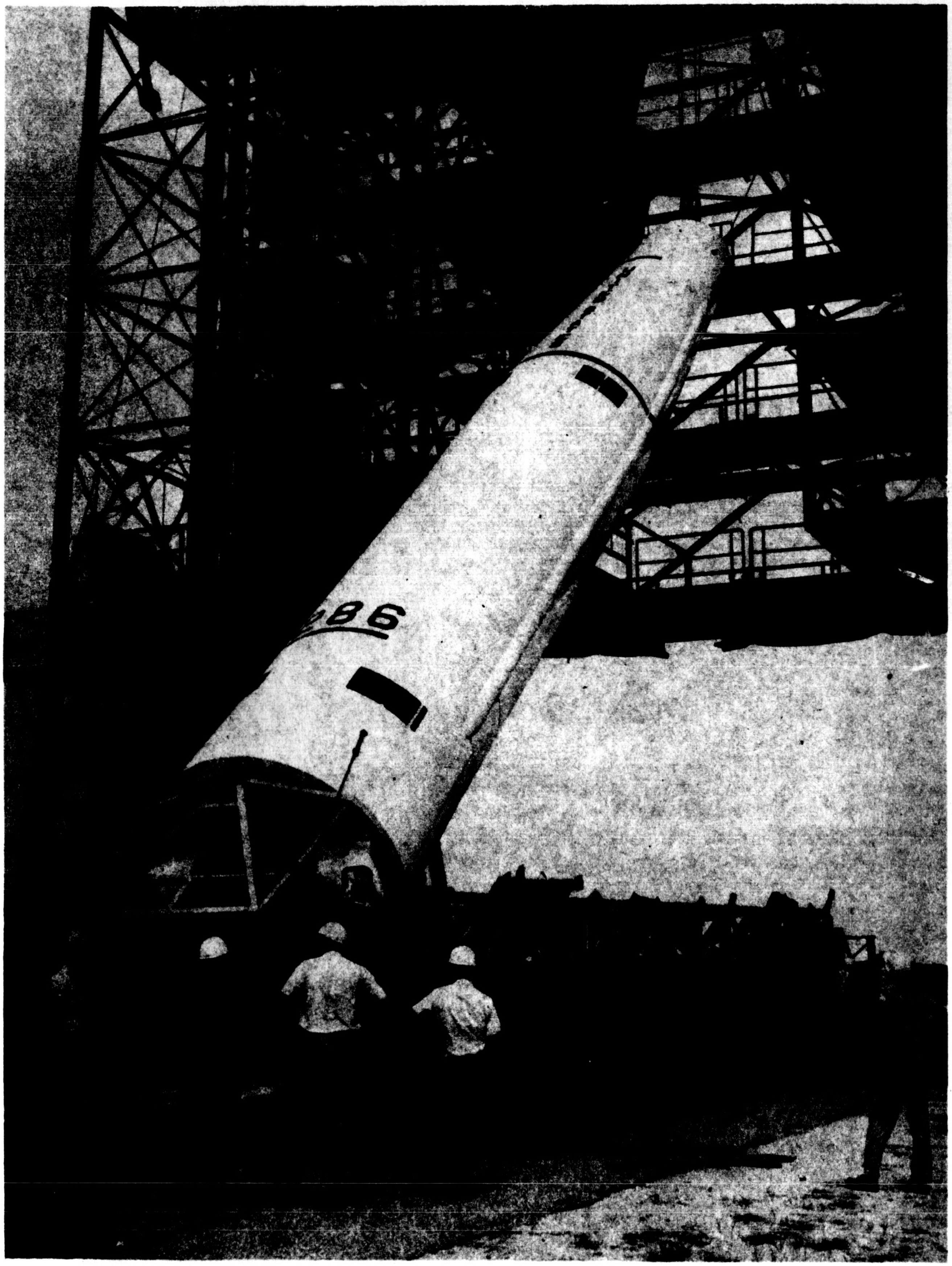
The main engine provides prime vehicle thrust and control during the first-stage powered flight phase of the vehicle trajectory. Directional control of the vehicle is accomplished by the gimballing of the main engine in response to signals from the flight controller. The two vernier engines provide both roll control as required during the first-stage powered flight, including programmed roll control for orientation of the vehicle in azimuth, and final thrust adjustments



DELTA DSV-3B OUTBOARD PROFILE
FIGURE 1



DELTA DSV-3B INBOARD PROFILE
FIGURE 2



ERECTION OF TYPICAL THOR DELTA FIRST STAGE

FIGURE 3

required to maintain a positive g force during second-stage ignition. The vernier engines can be gimballed in pitch and yaw in the same manner as the main engine. When the vernier engine hydraulic pitch actuators are activated differentially, the vernier engines move in opposite directions to produce vehicle roll response.

Performance characteristics of the first-stage propulsion system are given in Table I.

2.2 Second-Stage Vehicle

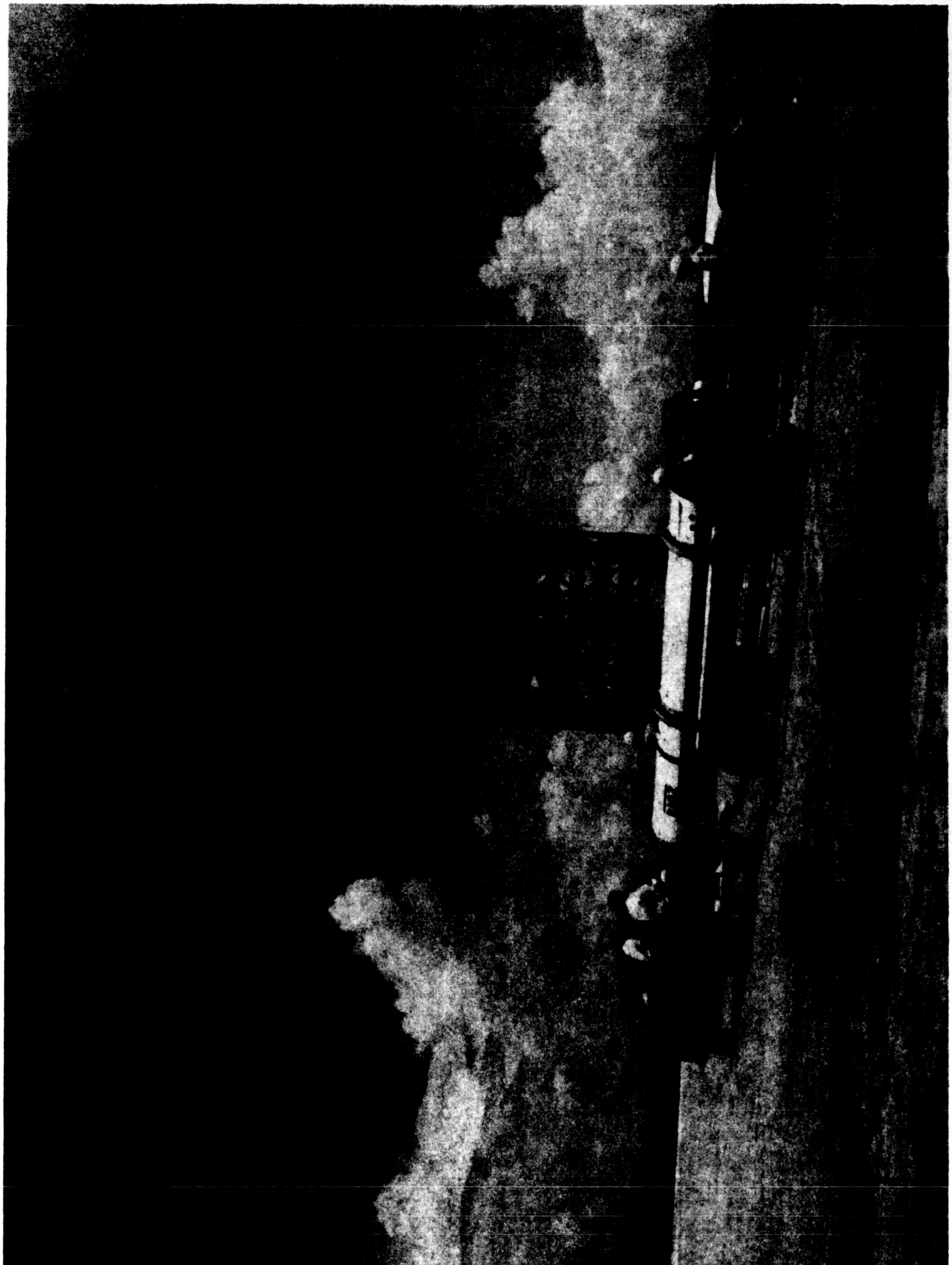
2.2.1 General Description

The second stage vehicle (figure 4) consists of an Aerojet General Corporation (AGC) AJ10-118A Propulsion System and an equipment and guidance compartment. The forward compartment houses the Bell Telephone Laboratories (BTL) guidance system, a flight controller, and the necessary instrumentation. A ball bearing assembly is mounted at the forward end of the equipment and guidance compartment. This ball bearing assembly supports the spin table, which accommodates spin stabilized payload requirements, and which also supports the third stage/payload combination.

2.2.2 Propulsion System

The second stage AJ10-118A Propulsion System consits of a regeneratively cooled thrust chamber assembly; propellant systems; helium pressurization system; a roll pitch and yaw attitude control system; and the interconnecting plumbing and airframe structure.

The thrust chamber is rated at 7,575 pound thrust (nominal) in vacuum; however, a gradual reduction in thrust level occurs during the latter part of burning due to a portion of the gas pressure supply being used for roll nozzle operation during this phase of operation, thereby reducing the propellant tank pressure.



TYPICAL THOR DELTA SECOND STAGE EN-ROUTE TO LAUNCH STAND

FIGURE 4

TABLE I

FIRST STAGE PROPULSION SYSTEM PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE
Thrust, Main Engine and Verniers (Nominal) Sea Level (30 second value sea level stabilized)	170,000 ± 2000
Altitude	193,400 lbs
Total Propellant Weight Flow (Nominal)	660 lb/sec
Mixture Ratio(Calibrated) at sea level WO/WF (Includes Re-circulation losses for Oxidizer)	2.15
Specific Impulse at Sea Level (Nominal)	250.4 sec
Main Engine Burning Time (Nominal)	150 sec
Vernier Engine Burning Time (To depletion)	163 sec
Propellant Utilization (Nominal)	99.5%
MIN. P.U. (3-sigma)	99%

Inhibited red fuming nitric acid and unsymmetrical dimethylhydrazine are utilized as oxidizer and fuel respectively. They are injected into the thrust chamber at a nominal mixture ratio of 2.76:1 by weight of oxidizer to fuel. The propellants are pressure fed to the thrust chamber by high pressure helium.

Performance characteristics of the second-stage propulsion system are given in Table II.

2.2.3 Guidance System

BTL radio command guidance steering is utilized during first and second stage powered flight. Second stage command cutoff is also accomplished by BTL.

In addition to the BTL system, programmers in the first and second stages provide coarse trajectory direction.

During powered flight, pitch and yaw control is accomplished by gimballing of the main thrust chamber assembly. Roll control is accomplished by expelling regulated high pressure gas through separate (clockwise and counterclockwise) pairs of nozzles during both powered flight and coast period durations. Subsequent to powered flight, pitch and yaw control can be accomplished, when desired, by expelling pressurized gas through nozzles installed in these two planes.

2.3 Third-Stage Vehicle

2.3.1 General Description

The third stage vehicle consists of an Allegany Ballistics Laboratories (ABL) X248-A5DM solid-propellant motor with forward and aft mounting rings. The motor is supported at the aft end of the cylindrical rocket motor case mounting ring by a hinged petal

TABLE II

SECOND STAGE PROPULSION SYSTEM PERFORMANCE CHARACTERISTICS

ITEM	PERFORMANCE
Thrust at Altitude (Nominal)	7,575 lbs
Total Impulse (Minimum)	1,250,000 lb-sec In Vacuum
Total Propellant Flow Rate (Nominal)	28.42 lb/sec
Mixture Ratio (Calibrated) at Altitude $\dot{W}_o/\dot{W}_f$ (Nominal)	2.76
Specific Impulse at Altitude (Nominal)	272 sec
Engine Burning Time (Nominal) For No Roll Nozzle Operation (Engine Burning Time and Thrust Decay Vary With Amount of Roll Nozzle Operation)	168 sec

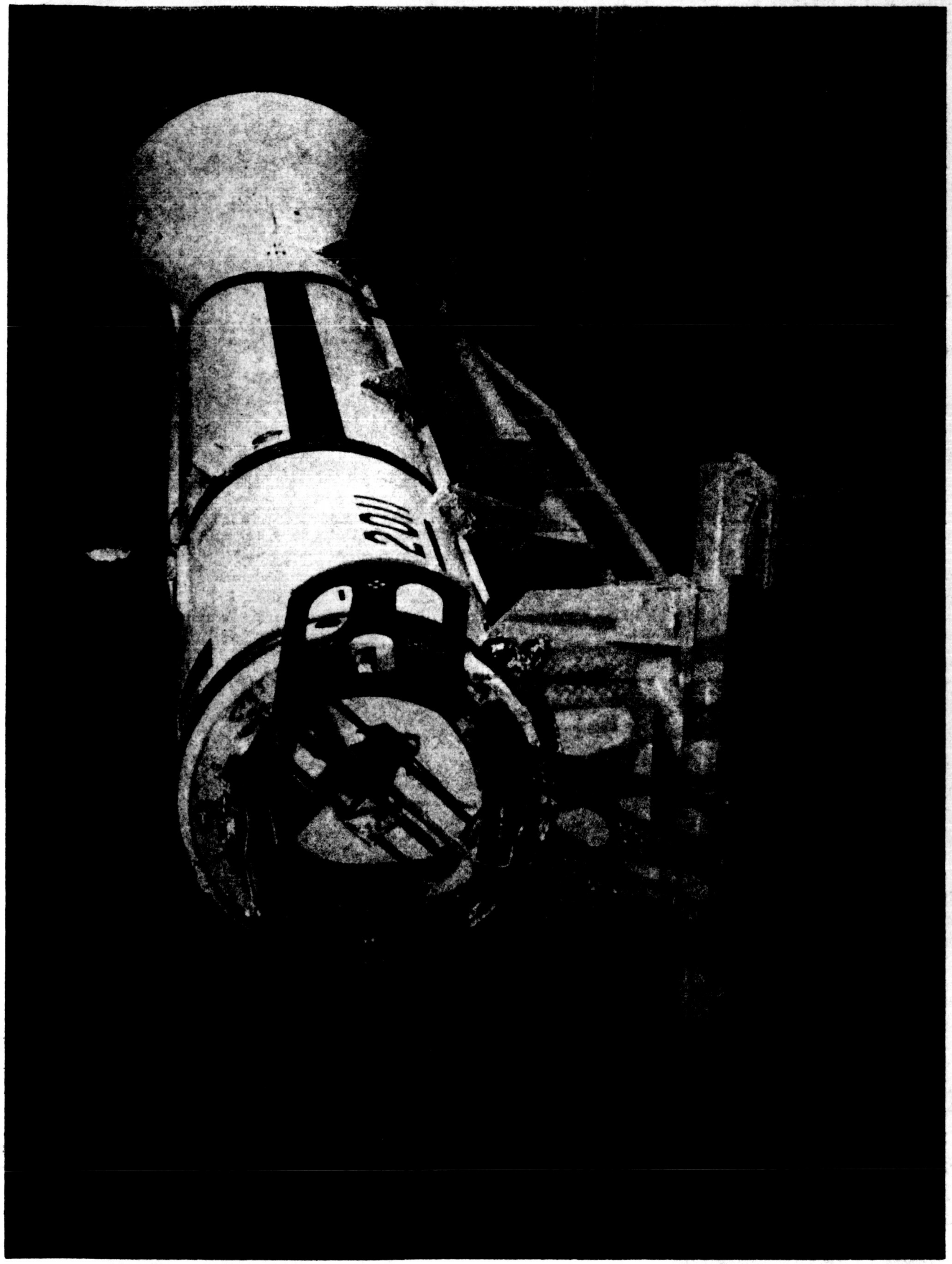
structure assembly (figures 5 and 6). This assembly includes a spin table and spin rockets. The aerodynamic fairing which covers both the payload and third stage (figure 7) is jettisoned during second stage operation. Spin-up of the third stage and payload is accomplished after second-stage burnout and an appropriate coast period. The third stage and payload is then released after the firing of explosive bolts which hold a band that secures the forward end of the petal support structure to the third stage motor.

The centrifugal force opens the petals allowing the spin stabilized third stage/payload combination to separate from the coasting second stage. At this same time, a reverse impulse is applied to the coasting second stage vehicle to provide the proper separation distance between the second and third stages. Third stage ignition then occurs approximately 15.5 seconds later. The support structure and spin table remain with the second stage vehicle.

Payload separation is accomplished by release of a restraining clamp, allowing a spring to separate the two objects.

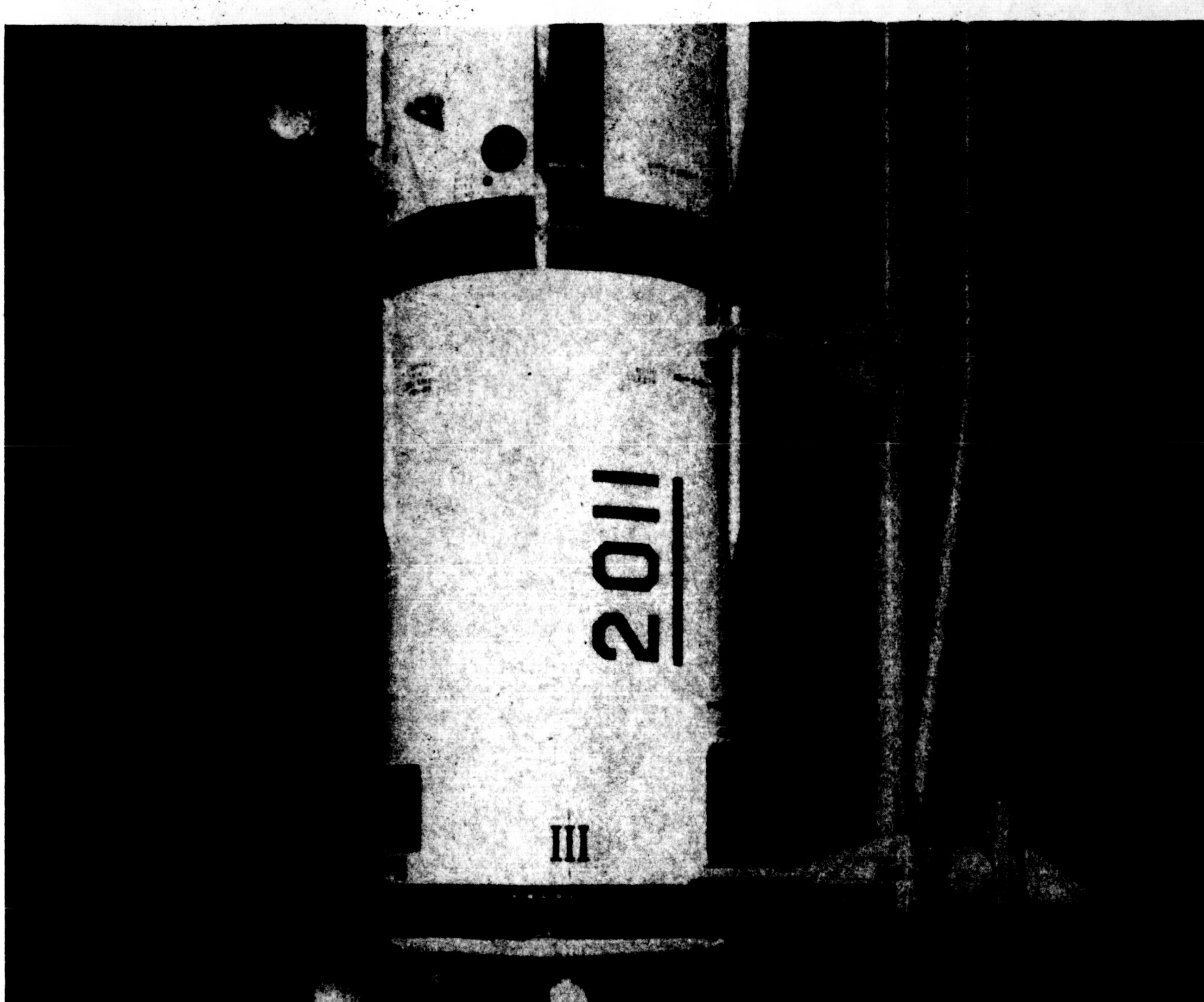
Approximately two seconds after payload separation, which occurs at least 120 seconds after third stage motor burnout, a YO system is deployed to tumble the third stage motor in order to prevent the third stage from impacting the payload.

Performance characteristics of the ABL X248 motor are presented in Table III.



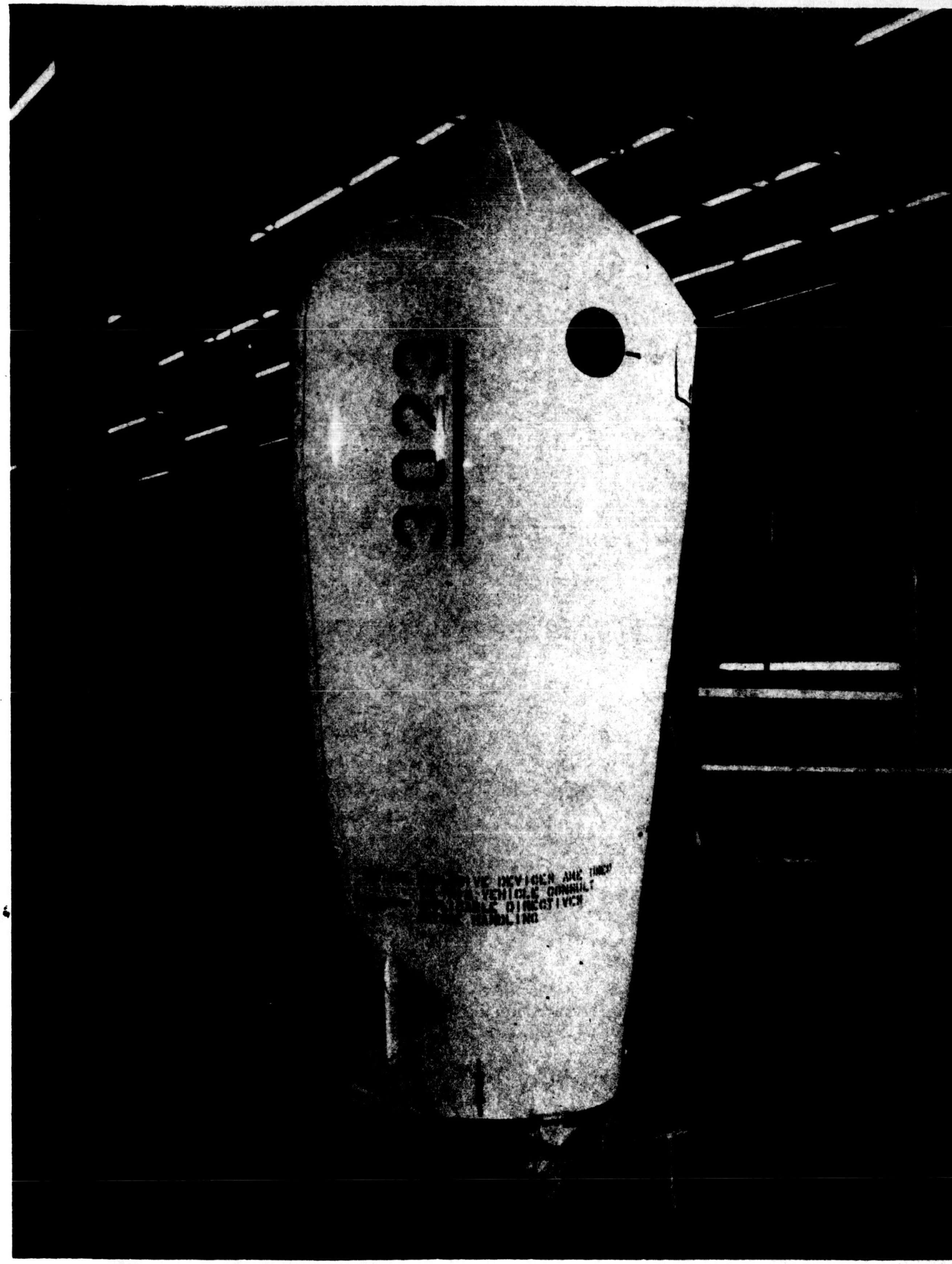
HINGED PETAL STRUCTURE ASSEMBLY (SPIN TABLE) FOR SUPPORT OF THIRD STAGE MOTOR

FIGURE 5



HINGED PETAL STRUCTURE ASSEMBLY SUPPORTING THIRD STAGE MOTOR

FIGURE 6



THOR/DELTA THIRD-STAGE BULBOUS FAIRING

FIGURE 7

TABLE III

THIRD STAGE PROPULSION SYSTEM PERFORMANCE CHARACTERISTICS AT 77°F

ITEM	PERFORMANCE
Average Thrust at Altitude (Nominal)	2,760 lbs
Maximum Thrust at Altitude	3,100 lbs
Consumed Weight (Based on Loss of 9 lbs of Inert Weight During Firing)	464.5 lbs
Specific Impulse in Vacuum (Nominal) (Based on Loss of 9 lbs of Inert Weight During Firing)	254.5 sec
Engine Burning Time	42 sec
Total Impulse at Altitude	115,875 lb-sec

3.0 THOR/DELTA DSV-3B PAYLOAD CAPABILITIES

The polar orbit capabilities of the Delta DSV-3B space vehicle with a low drag fairing have been determined. In the analysis, considerations of range safety, aerothermodynamic heating, and payload requirements (i.e. spin axis orientation) were omitted. The first and second stages were programmed to follow a ballistic trajectory (zero angle of attack) after an early first stage pitch rate of short duration. The payload was injected into orbit by the third stage at second stage apogee.

Typical instantaneous impact point (IIP) traces for southward launches from the Pacific Missile Range are shown in figure 8. Shown in this figure are nominal stage 1 (MECO), Stage II (SECO) and stage III propellant depletion shutdown (PDS) impact points. Also shown are typical three sigma lateral dispersions and impact ellipses assuming the guidance system inoperative.

The payload weight capabilities for polar orbits are shown in figures 9 and 10. Figure 9 presents elliptic and circular orbit capabilities while figure 10 presents the space probe capabilities. The first and second stage nominal propellant utilization (PU) are 99.6 and 96 percent respectively. Payload support equipment was assumed to be 15 pounds. Launch was assumed to be at the latitude of the Pacific Missile Range; however, for polar orbits launch latitude has a very small effect on payload capability. Figures 9 and 10 are therefore general.

Impact surface range at MECO, SECO and Stage II PDS are shown in figure 11 as a function of Stage II apogee altitude. MECO surface range is very insensitive to payload weight. The semi major axis of the Stage II impact ellipse in the range direction is very sensitive to trajectory shaping varying between 200 to 500 nautical miles for medium altitude orbits up to 1500 nautical miles for low second stage apogee space probes. Note that in order to avoid impacts on land (for specific launch sites) Stage II impact point can be subranged

DELTA DSV-3B
PERFORMANCE CAPABILITIES
IIP FOR TYPICAL TRAJECTORY AND FOR TYPICAL THREE SIGMA DISPERSIONS

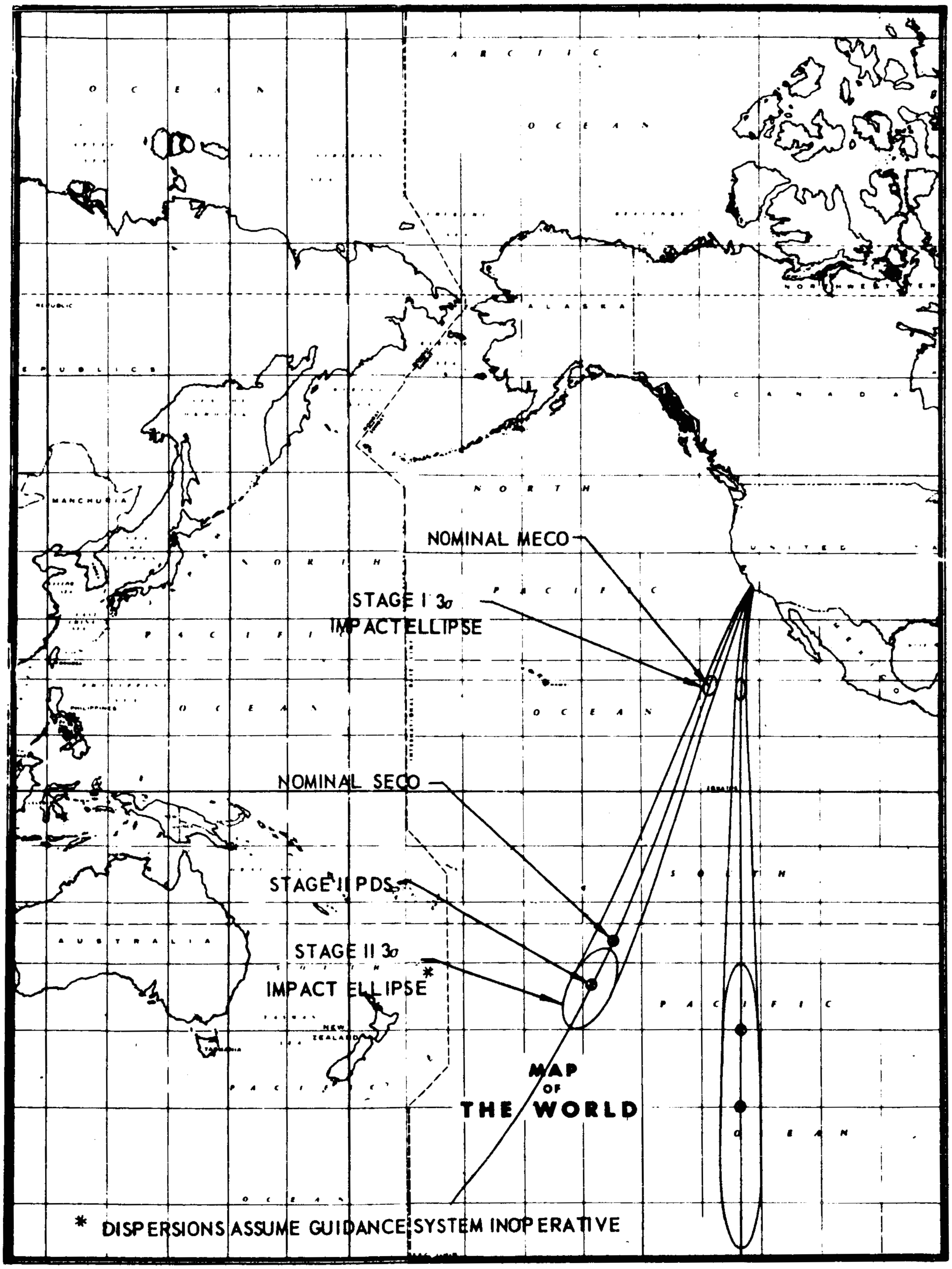
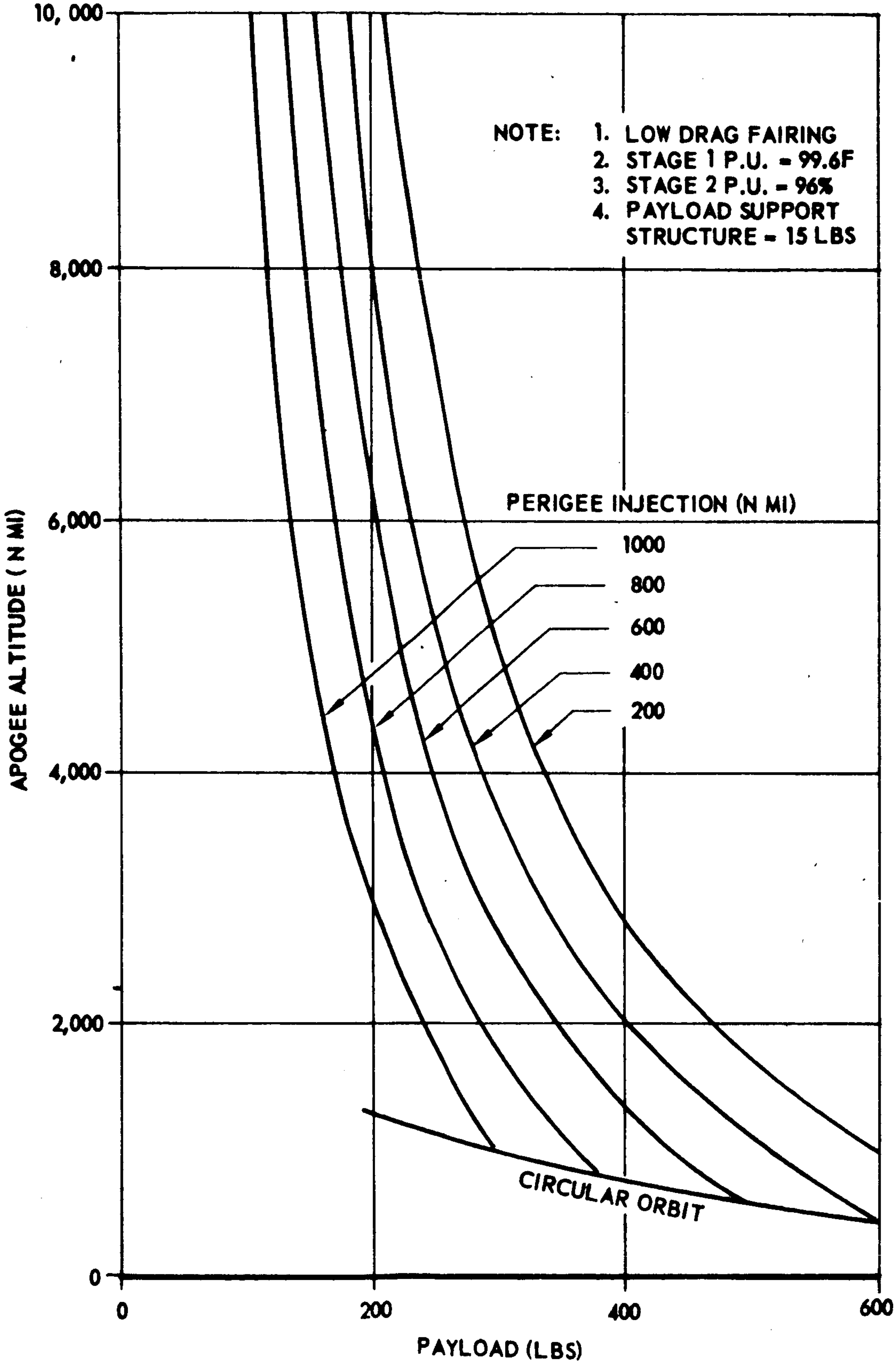


FIGURE 8

DELTA DSV-3B
PERFORMANCE CAPABILITIES
ELLIPTIC ORBIT CAPABILITY POLAR ORBIT



DELTA DSV-3B
PERFORMANCE CAPABILITIES
SPACE PROBE CAPABILITY POLAR ORBIT

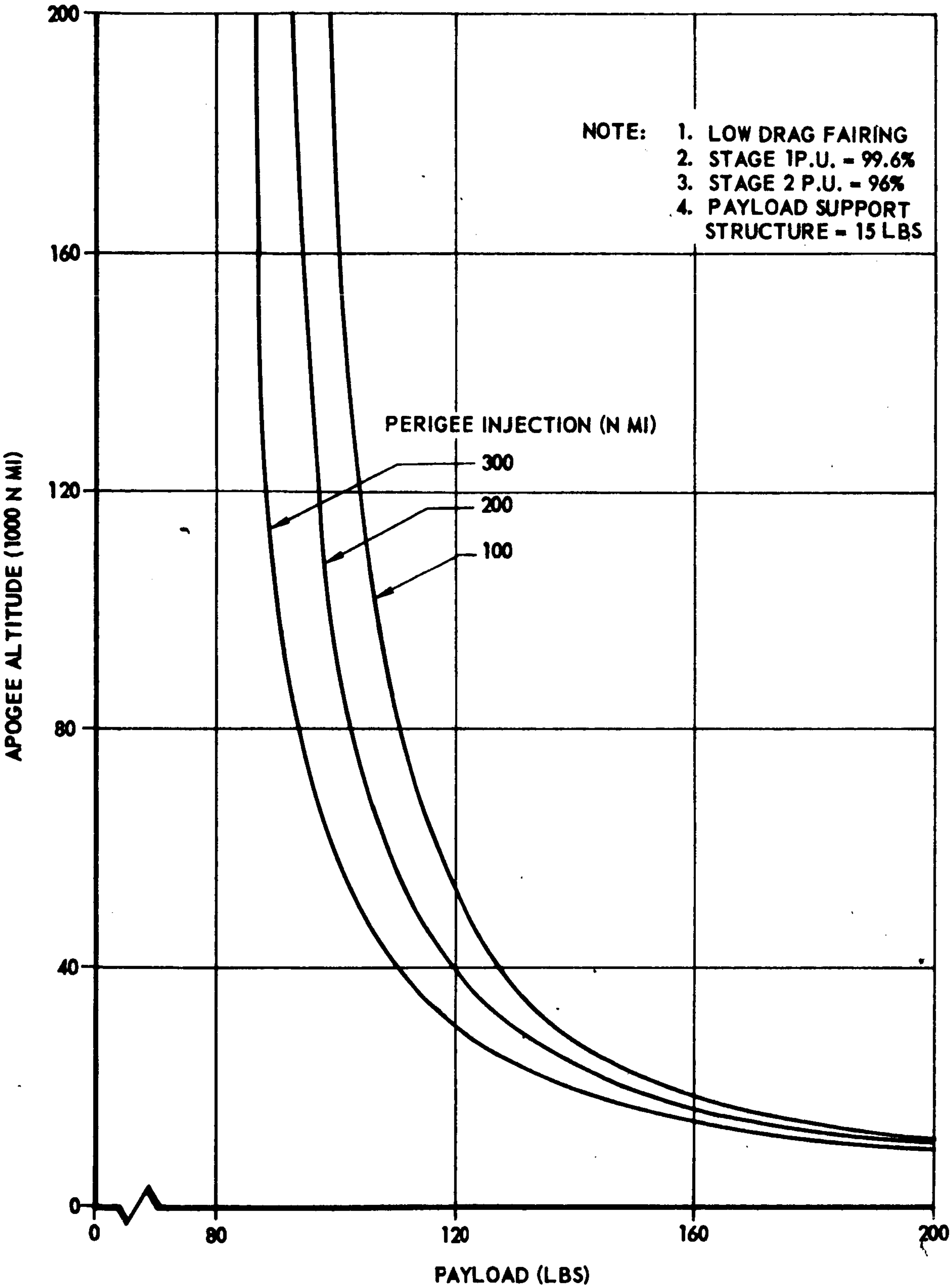
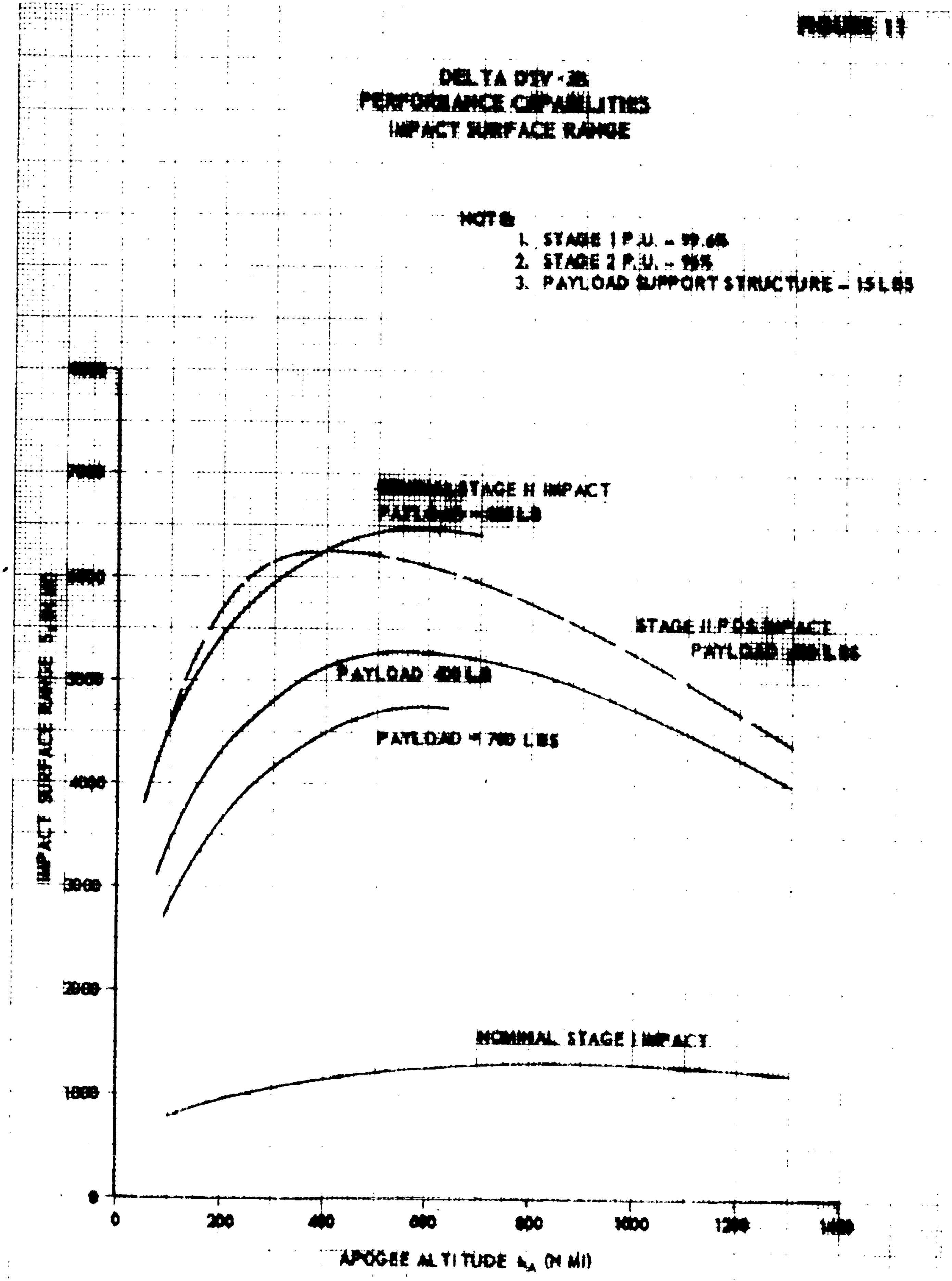


FIGURE 11


DELTA DSV-3E
PERFORMANCE CAPABILITIES
IMPACT SURFACE RANGE
NOTES:
1. STAGE I P.U. – 99.6%
2. STAGE 2 P.U. – 96%
3. PAYLOAD SUPPORT STRUCTURE – 15 LBS
STAGE II IMPACT
STAGE II P.D.S. IMPACT
PAYLOAD 400 LB
PAYLOAD = 700 LBS
NOMINAL STAGE I IMPACT
IMPACT SURFACE RANGE
APOGEE ALTITUDE h_A (N MI)
0
1000
2000
3000
4000
5000
6000
7000
0
200
400
600
800
1000
1200
1400

and to a certain extent extended by proper trajectory shaping and at the expense of payload weight.

In addition, for certain launch sites and inclinations, a dog-leg maneuver in which either the second stage or third stage or both are yawed, must be performed in order to avoid land overfly. Third stage yaw angles up to about 30 degrees were investigated for 600 and 1000 nautical mile circular orbits. The payload losses as a function of change in payload injection velocity azimuth are shown in figure 12. As seen, the weight loss for a given change in azimuth is less for lighter payloads.